File D 52.1 / U. S. B. 1.A / 5

McCOOK FIELD REPORT, SERIAL No. 1815

AIR SERVICE INFORMATION CIRCULAR

(AVIATION)

PUBLISHED BY THE CHIEF OF AIR SERVICE, WASHINGTON, D. C.

Vol. IV | May 1, 1922 | No. 338

REPORT OF STATIC TEST OF XB-1-A FUSELAGE

(AIRPLANE SECTION, S. & A. BRANCH)

Prepared by Engineering Division, Air Service
McCook Field, Dayton, Ohio
January 4, 1922

WASHINGTON
GOVERNMENT PRINTING OFFICE
1922

CERTIFICATE: By direction of the Secretary of War, the matter contained herein is published as administrative information and is required for the proper transaction of the public business.

REPORT OF STATIC TEST OF XB-1-A FUSELAGE.

OBJECT.

The object of this test was to determine the strength of an XB-1-A fuselage which had seen 83 hours 25 minutes service in the air and been subjected to weather conditions for 1 year 4 months and 4 days from the time of assembly until the airplane was condemned. The time of service was from July 1, 1919, to November 4, 1920.

DATE AND PLACE.

The test was conducted February 23, 1921, at McCook Field, Dayton, Ohio.

WITNESSES.

Maj. E. L. Napier. Mr. W. E. Savage.
Lieut. E. W. Dichman. Mr. D. B. Weaver.

SUMMARY.

The required load factor for a fuselage of this type is 7. At a load factor of 3½ the veneer skin buckled between points C and D. At a load factor of 5.5 the veneer skin cracked open at the rear supports in the jig. At load factors of 6.5 and 7, respectively, the bottom of the fuselage buckled between No. 4 and No. 5 bulkheads back of the rear landing-gear fittings.

FUSELAGE.

DESCRIPTION.

This fuselage was built by J. C. Widman & Co., of Detroit, Mich. The overall dimensions and the general design were the same as in the other XB-1-A fuselages tested.

Figure 1 shows plan and side views of the fuselage.

The longerons of this fuselage were of solid gray elm, while the bulkheads and skin were laminated.

The laminations of the bulkheads alternate with elm and birch, with a core of either elm, poplar, or birch.

Figure 2 is a drawing of bulkhead No. 7.

The thickness and plies are as tabulated below:

Bulkhead number:

1, $\frac{7}{8}$ inch thick, 13 ply, $12\frac{1}{16}$ inch, with $\frac{1}{8}$-inch core.
2, $\frac{7}{8}$ inch thick, 13 ply, $12\frac{1}{16}$ inch, with $\frac{1}{8}$-inch core.
3, $\frac{7}{8}$ inch thick, 13 ply, $12\frac{1}{16}$ inch, with $\frac{1}{8}$-inch core.
4, $1\frac{1}{8}$ inch thick, 13 ply, $13\frac{1}{16}$ inch.
5, $\frac{9}{16}$ inch thick, 9 ply, $9\frac{1}{16}$ inch.
6, $1\frac{1}{8}$ inch thick, 13 ply, $13\frac{1}{16}$ inch.
7, $\frac{3}{4}$ inch thick, 12 ply, $10\frac{1}{16}$ inch, with $\frac{1}{8}$-inch core.
8, $\frac{1}{2}$ inch thick, 7 ply, $6\frac{1}{16}$ inch, with $\frac{1}{8}$-inch core.
9, $\frac{9}{32}$ inch thick, 9 ply, $9\frac{1}{32}$ inch.

All remaining bulkheads were the same as No. 9.

Bulkheads 4, 6, 7, and 8 were reinforced in thickness where fastened to longerons and other members.

The gluing area for attaching the skin to the bulkheads from No. 8, inclusive, to the rear was increased by adding a strip ¼ inch thick by ½ inch wide to rear side. The sections of the skin were joined together by glued scarf joints and ½-inch No. 4 flathead brass screws spaced 2 inches center to center.

The total weight of this fuselage was 210 pounds.

The condition of the fuselage when received for test was as follows:

The paint on the engine bearers was blistered with heat as seen in figure 5.

No. 3 bulkhead was pulled loose from the longerons on the lower right side.

No. 3 bulkhead was pulled away from the plywood covering on both sides and bottom, due to shocks received in landing.

The nails were pulled on No. 3 bulkhead on the right side, as shown in figure 6.

The plywood skin was buckled on the left side, as shown in figure 7.

All bulkheads from the front of the fuselage as far back as the rear of the gunner's cockpit were pulled loose from the longerons.

The whole bottom of the fuselage was covered with oil.

Figure 8 shows the condition of the bottom of the fuselage.

PROCEDURE FOR TEST.

The fuselage was mounted in a jig and supported by the wing fittings.

The loading was distributed and applied as indicated in the loading schedule in figure 3.

The tail load was carried by a platform suspended on a cable which transferred the load to the fuselage proper.

After each increment of the load had been put on, the jacks under this platform were released, so as to allow the load to act on the rear part of the fuselage structure.

RESULTS.

The deflections in inches and the tabulated results may be seen in figure 4.

At a load factor of 3.5 the veneer skin buckled at a point just to the rear of the rear wing fittings.

When a factor of 5.5 was reached the veneer cracked on both sides at the rear wing fittings.

At a load factor of 6.5 and 7 the veneer skin buckled on the bottom between bulkheads 4 and 5 back of last landing gear fittings.

Complete failure occurred at a load factor of 7.5.

Figure 9 is a photo of the failure.

97990—22

DISCUSSION.

All the other tests on the XB–1–A fuselage were made on new specimens, and the failures occurred at factors as tabulated below:

No. 1—Davies-Putnam fuselage (Bristol) XB–1–A.—Tested July 29, 1918; failed at a load factor of 7.5.

No. 2—Davies-Putnam fuselage (Bristol) XB–1–A.—Tested July 30, 1918; failed at a load of 7.5.

No. 3—Davies-Putnam fuselage (Bristol) XB–1–A.—Tested August 7, 1918; failed at a load factor of 7.

No. 1—J. C. Widman & Co. fuselage (Bristol) XB–1–A.—Tested July 31, 1918; failed at a load factor of 7.

The fuselage mentioned in this test had 1 year 4 months and 4 days' service and supported a loading equivalent to a load factor of 7.

It required a load factor of 7.5 to bring about a complete failure.

CONCLUSION.

The second J. C. Widman & Co. XB–1–A fuselage was entirely satisfactory structurally.

Since this fuselage had over 1½ years' service without any indications of weakness, it is apparent that the life of the fuselage is much longer than this. With good care there is every reason to suppose that the life of this particular fuselage will be about 3 or 4 years.

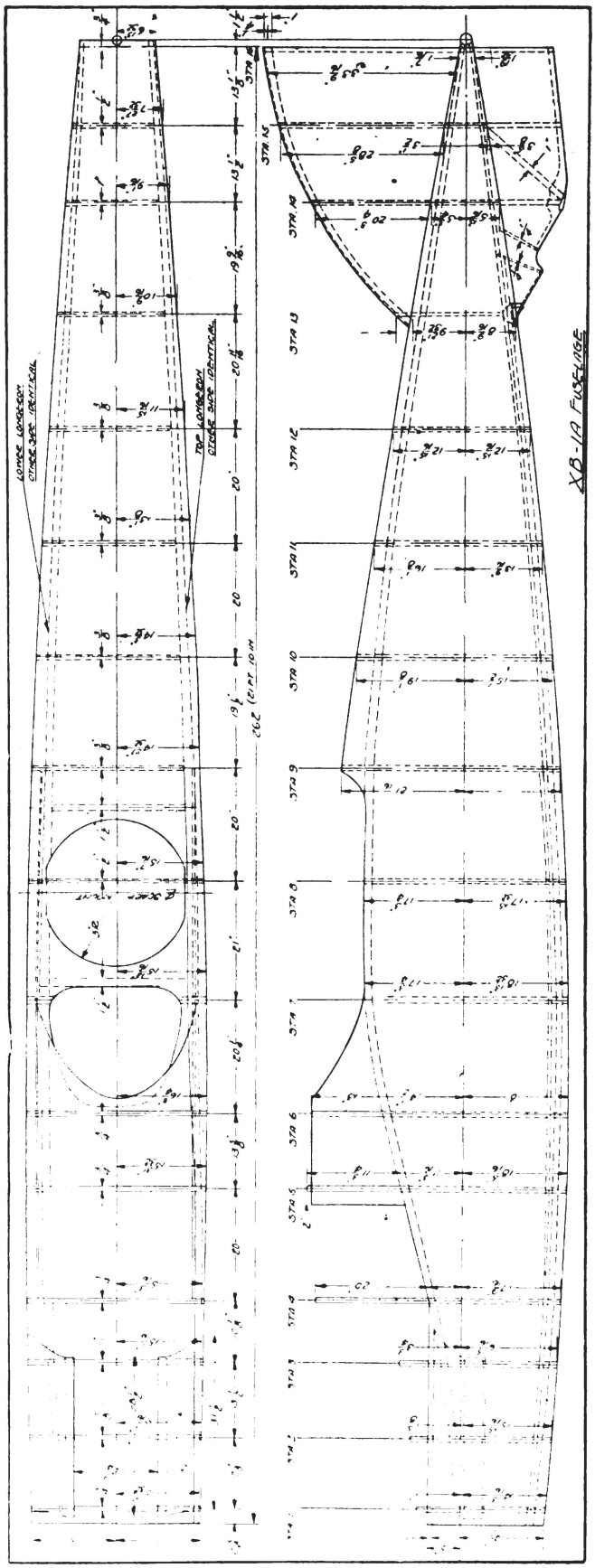


FIG. 1.

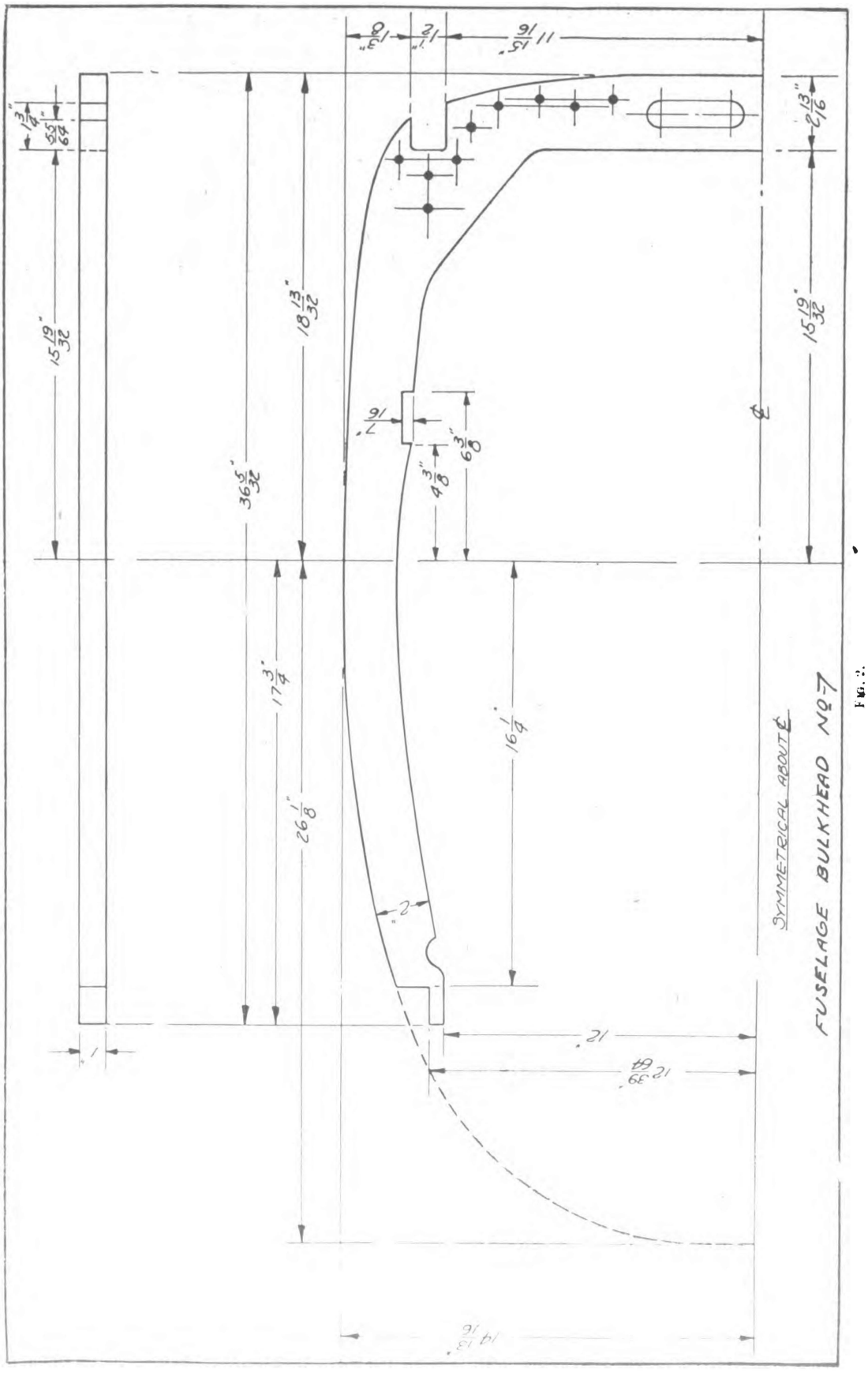
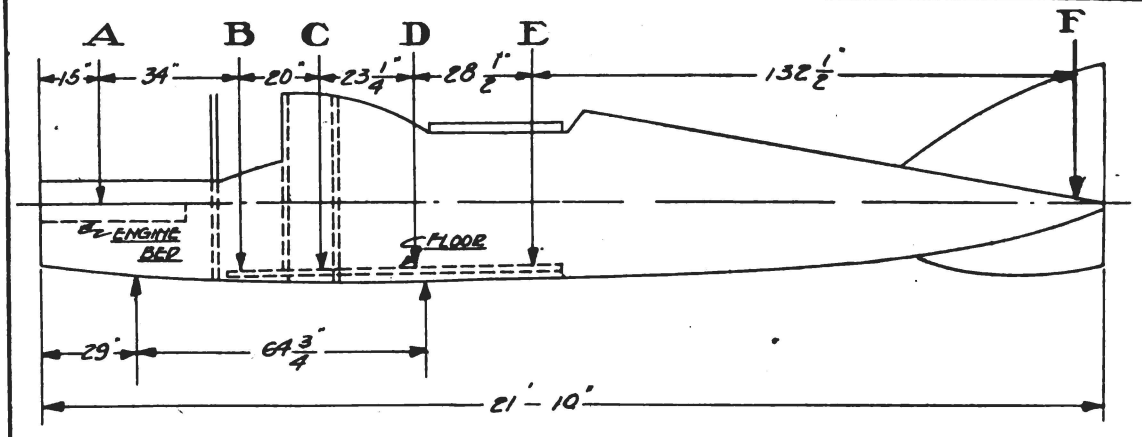


Fig. 3.

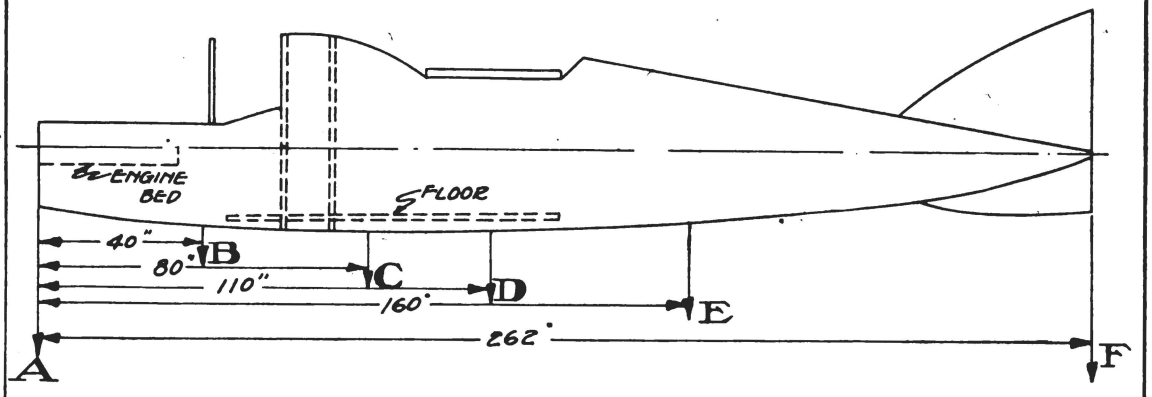


FUSELAGE LOADING SCHEDULE

LOAD NO	DYNAMIC FACTOR	TAIL LOAD LBS. PER SQ. FT.	LOAD IN LBS. AT						TOTAL LOAD ON FUSELAGE
			A	B	C	D	E	F	
1	2	10	1660	1356	280	560	640	604	5098
2	2.5	12.5	2075	1695	350	700	800	755	6373
3	3	15	2490	2034	420	840	960	906	7648
4	3.5	17.5	2905	2373	490	980	1120	1057	8923
5	4	20	3320	2712	560	1120	1280	1208	10,198
6	4.5	22.5	3735	3051	630	1260	1440	1362	11,473
7	5	25	4150	3390	700	1400	1600	1510	12,748
8	5.5	27.5	4565	3729	770	1540	1760	1661	14,023
9	6	30	4980	4068	840	1680	1920	1812	15,298
10	6.5	32.5	5395	4407	910	1820	2080	1963	16,573
11	7	35.0	5810	4746	980	1960	2240	2114	17,848
12	7.5	37.5	6225	5085	1050	2100	2400	2265	19,125

XB-1A FUSELAGE

FIG. 3.



LOAD FACTOR	DEFLECTION IN INCHES AT						REMARKS
	A	B	C	D	E	F	
2	1.6	1.4	.8	.2	-.4	-.6	
2.5	1.7	1.5	.9	.3	-.2	-.5	
3	1.8	1.6	1.0	.5	0	-.2	
3.5	1.8	1.7	1.0	.5	+.1	.2	BUCKLE IN VENEER TO REAR OF REAR WING FITTING.
4	1.9	1.7	1.1	.7	.5	.5	
4.5	1.9	1.9	1.2	.9	.6	.9	
5	2.0	1.9	1.3	1.0	.8	1.4	
5.5	2.0	1.9	1.4	1.2	1.1	1.7	VENEER CRACKED OPEN BY BUCKLING AT REAR WING FITTINGS.
6	2.1	2.0	1.4	1.4	1.3	2.2	
6.5							BUCKLING IN BOTTOM BETWEEN No. 4 & 5 BULKHEADS, JUST BACK OF
7							LANDING GEAR FITTING.
7.5							FAILURE

XB-1A FUSELAGE

FIG. 4.

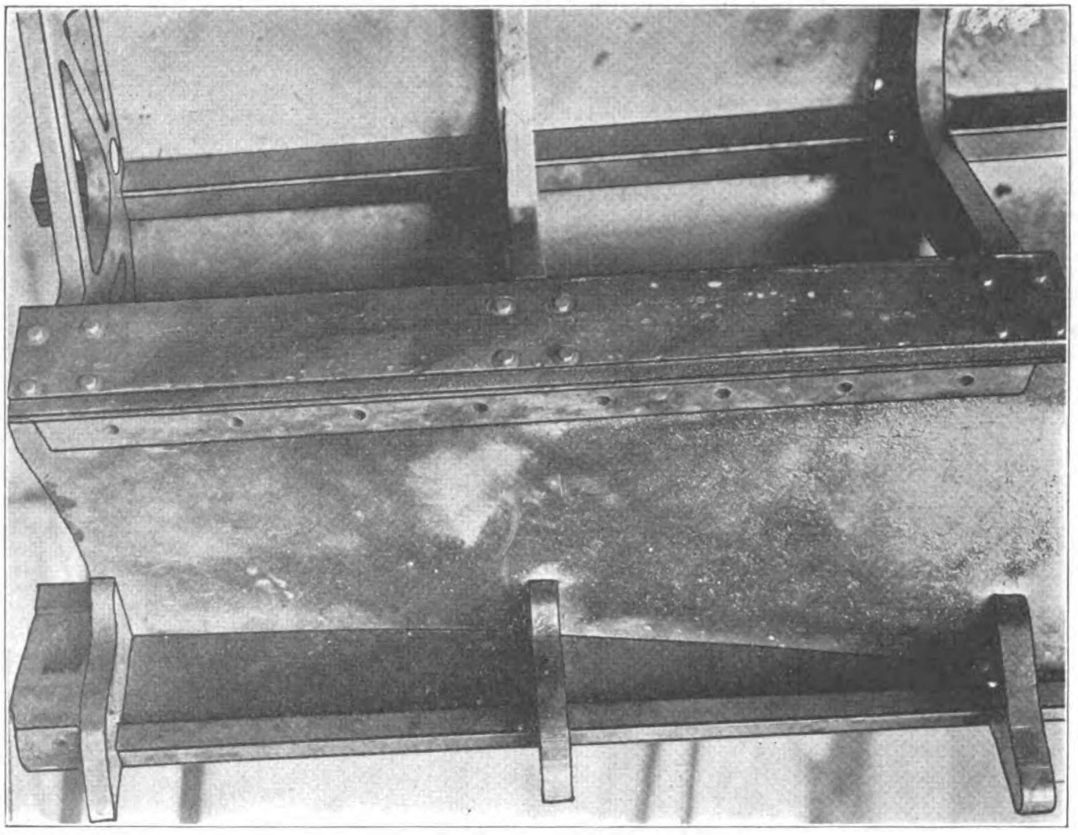


FIG. 5.

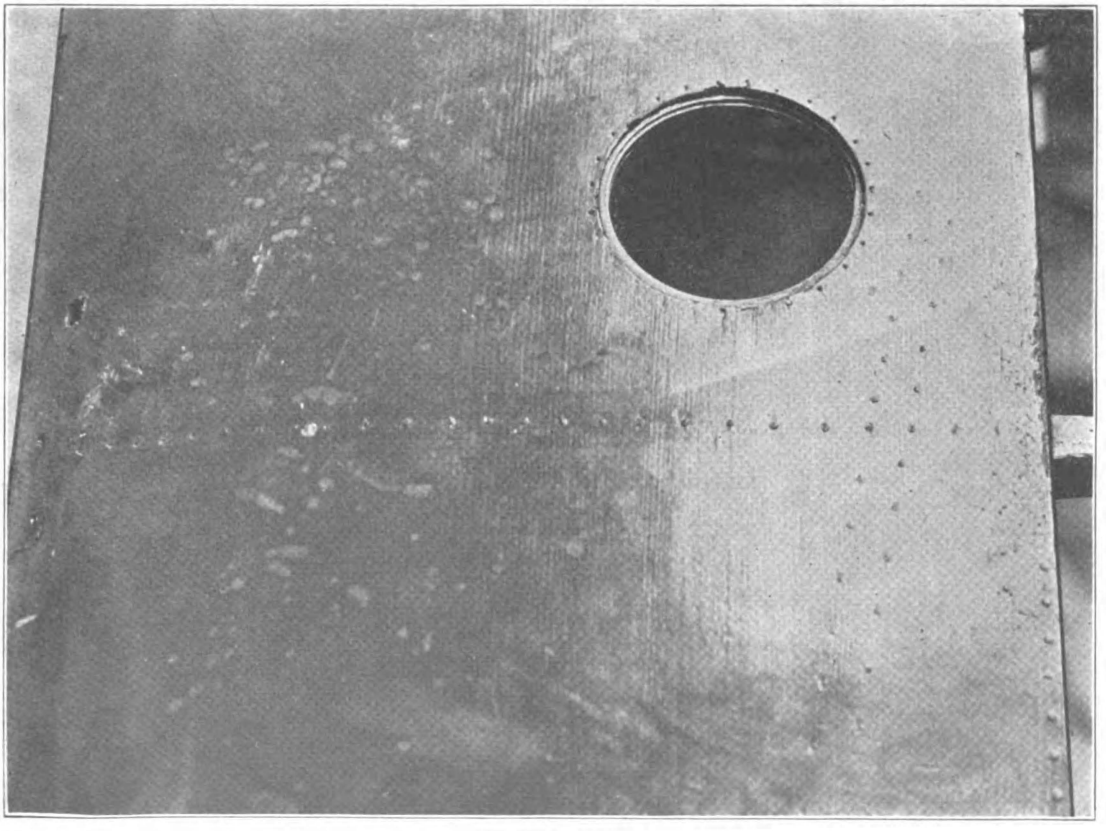


FIG. 6.

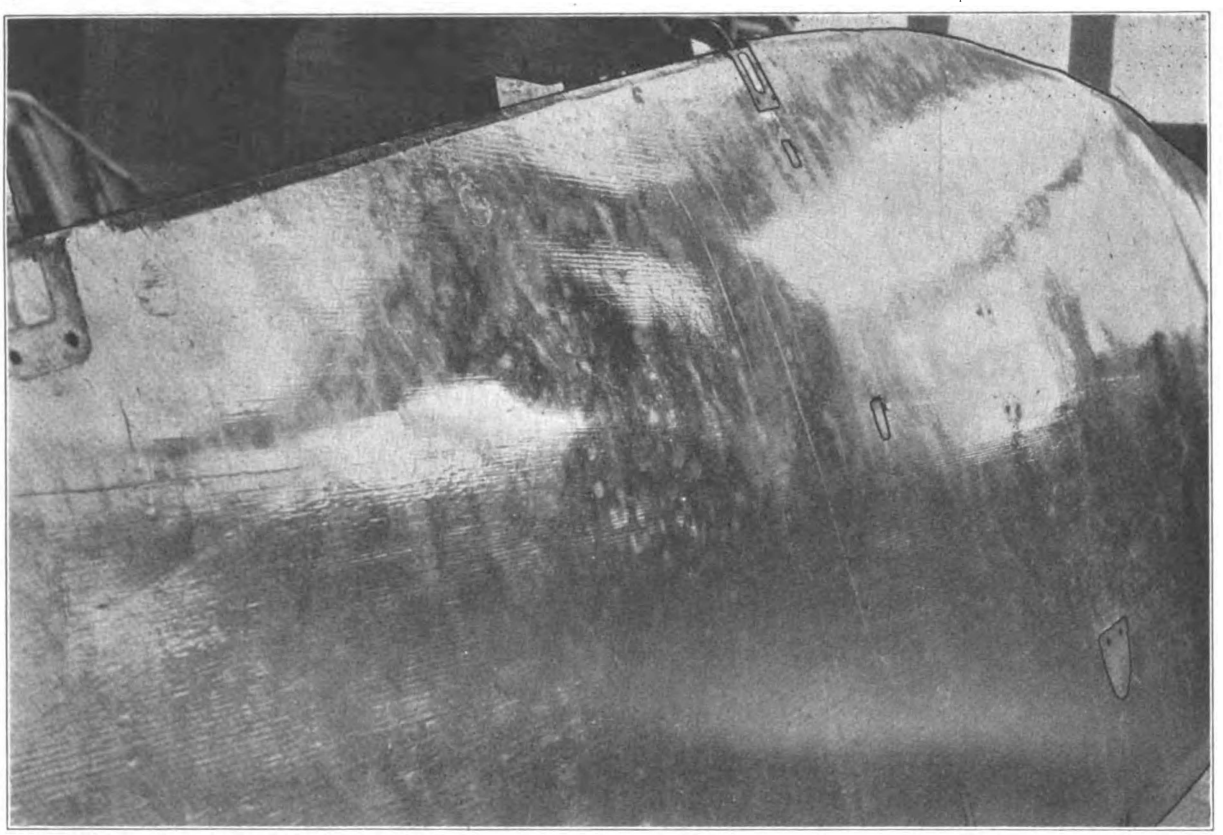


FIG. 7.

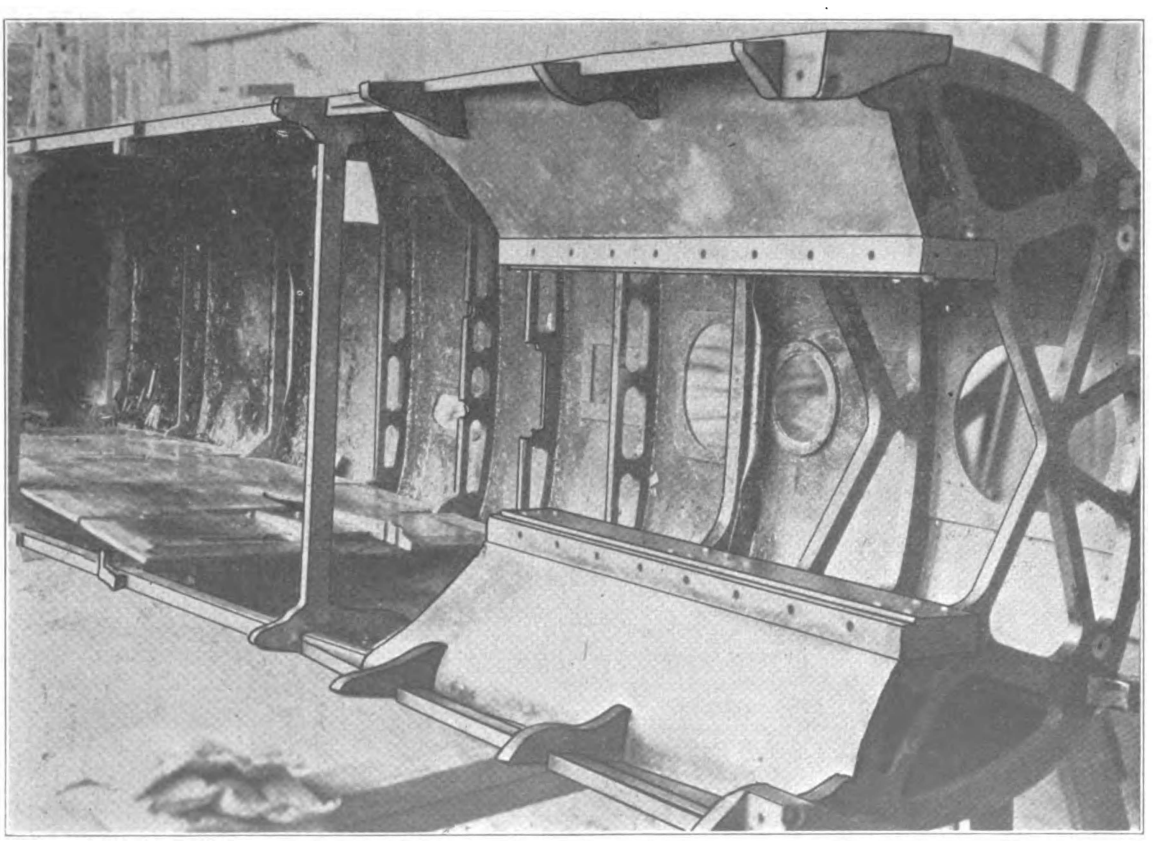


FIG. 8.

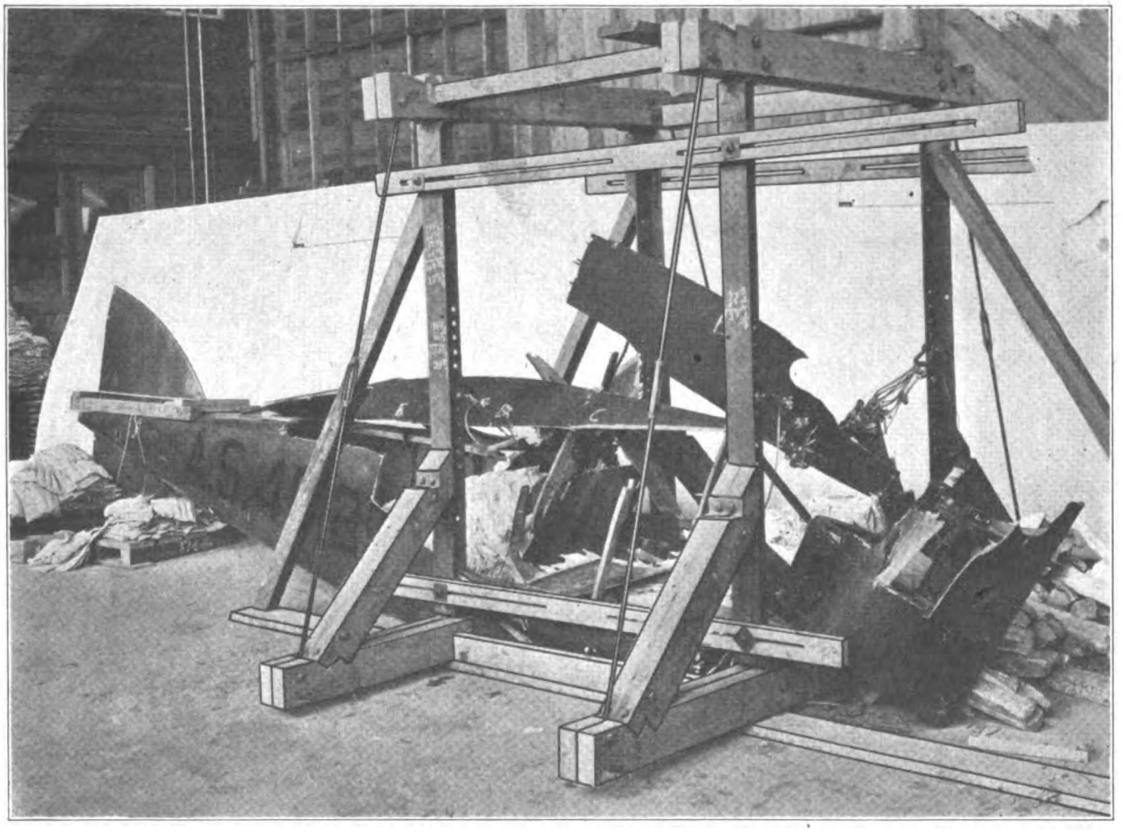


FIG. 9.